COST-EFFECTIVE MISSION DESIGN FOR A SMALL SOLAR PROBE

by

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ABSTRACT

The Small Solar Probe Mission is designed to characterize the Solar Corona as close as three solar radii above the sun's surface. Until 1992 studies generated concepts with large spacecraft launched on large launch vehicles and requiring nearly a decade to reach their initial perihelion. In 1992 JPL responded to emerging fiscal constraints by proposing a Small Solar Probe. The payload was reduced in scope while yet retaining the primary objectives of this exploratory mission. The mission design uses a Delta launch vehicle and a direct trajectory to Jupiter where a gravity assist places the spacecraft on a polar trajectory about the Sun. This paper describes the mission and the designs derived from it in the context of a cost effective concept. Dosign choices made are directly related to cost implications.

INTRODUCTION

One of the last unexplored regions of the solar system is that within 0.3 AU of the Sun. The goal of the Solar Probe mission is to go to within three solar radii of the Sun's surface. 1 his will enable *in situ* observations of phenomena related to the origin and acceleration of the solar wind, the energy balance within the corona, and the plasma and dust to be found there.

Until 1992 all concepts for this mission consisted of large spacecraft with payloads exceeding 100 kg. Even though these designs would be launched on large launch vehicles they still required trajectories with long flight times. In this way the required launch energies and/or the required post-launch maneuver budgets could be accommodated in the flight system design. When considering life-cycle costs the potential existed for the operations budget to push the concept above acceptable limits.

In February 1992 the Jet Propulsion Laboratory responded to the current fiscal environment by proposing to the Space Physics Division, at NASA Headquarters, that a Small Solar Probe (SSP) concept be developed. It would weigh at most a few hundreds of kilograms and would be able to be launched on a small launch vehicle (such as a Delta). In addition the flight time should be constrained to be of the order of three years to perihelion.

The science community has responded by developing a strawrnan payload that only weighs 20.4 kg and requires 18.2 W of power. Mission designers have recognized the benefits associated with a direct trajectory to Jupiter and thence to the

Sun. System designers have responded by developing conceptual designs that weigh no more than 170 kg, a mass that is consistent with such trajectories and with the capacity of the Delta launch vehicle. This paper describes the cost effectiveness of such a design.

SCIENCE INVESTIGATIONS

In response to NASA's desire for missions that are better, cheaper and faster and those that also use new technology, the NASA Division of Space Physics sponsored a Workshop where smaller, lightweight space plasma instrument concepts were presented and discussed. We have incorporated these ideas in a new concept of Solar Probe called the Small Solar Probe. The new mission focusses on the purely exploratory region, that between 0.3 AU and 4 R_s from the center of the Sun. Because of the high magnetic field, photon and X-ray intensities, and plasma and energetic particle fluxes in this region, it is possible to achieve all of the major science goals with smaller instruments. The reduced spacecraft payload weight and power requirements allow a much smaller supporting spacecraft and launch vehicle, thus reducing overall costs.

Table 1 shows the Small Solar Probe Science payload. This payload was selected by the NASA Small Solar Probe Science Study Team (SST) during a meeting in March 1993. These instruments are: a plasma detector, a magnetometer, a plasma wave detector, an energetic ion experiment, a coronal photometer, a hard X-ray photometer, and two shared DPUS (the spacecraft has shared power supplies for all systems). The total weight is 20.4 kg and the power is 18.2 Watts, including 20% contingency.

Table 2 is the SSP SS1 augmented payload. Additional capabilities of the augmented payload are: the plasma detector offers ion composition measurements, and a 2 π steradian FOV; the plasma wave instrument will make 3-axis and broadband wave measurements; the magnetometer has a longer boom; the energetic ion instrument will include composition measurements in the suprathermal range and a stepping motor. The coronal photometer now provides scanning, The X-ray photometer includes a xenon detector. 1 his augmented payload of 34.3 kg and 32.7 Watts (including contingency) is feasible for the 55P. Currently, two industrial contractors are examining the spacecraft concept in detail which will allow us to determine more accurately the maximum allowable payload.

1 he science that SSP will achieve is given in Figure 1. This is basically the same science as the full Solar Probe mission with a reduction in mission cost by a factor of ~ 5 and a much shorter mission lifetime ~ 1/2. 1 he shorter lifetime reduces costs significantly and increases reliability. I or a full discussion of the < 0.5 AU science objectives, we refer the reader to Solar Probe Scientific Rationale and Mission Concept[2].

Figure 1: Small Solar Probe Science Objectives

- Investigate Heliosphere Inside Helios Distance Of 0.3 AU (60 R_S) - A Region Never Previously Explored
- 30-4 R_S, Investigate: The Nature, Origin, And Effects Of Plasma Instabilities
  Solar Wind Angular Momentum Transport
  The Alfven Critical Point
- 10-4 R_S, Investigate: Distance Where Solar Wind Becomes Supersonic
  Evidence Of Solar Cosmic Ray Acceleration And Release
- Perihelion OF 4 R_S Planned:
  Major Portion Of Solar Wind Acceleration Occurs From 2-8 R_S (Nominal Sonic Point ~4 R_S)
  Must Be As Close As Possible To Detect Closed Magnetic Field Lines
  2-10 R_S Particle Storage
- Also Study Coronal Mass Ejections And Microflares (Shocks And Particle Acceleration)
- Investigate Properties Of Interplanetary Dust (Are There Circumsolar Rings?)

Table 1: Small Solar Probe Minimum Payload

Instrument	Mass (kg)	Power (W)	Bit Rate (kbps)	Comments
Plasma Detector	3.7	2.7	1.2	Ions, electrons, TOF, π ster FOV
Magnetometer	2.5*	1.5	0,5	*Includes 0.5 kg boom
Plasma Wave	4.25	4	0,5	30 Hz to 30 MHZ, 1-axis, includes 0.25 kg for antenna
Energetic Ions	2.5	2.5	0.5	$0.02 < E_e < 1$ MeV, $0.02 < E_p < 10$ MeV G_{ions}=0. 1, $G_{electrons}$= 0.002 cm^2ster
Coronal Photometer	0.75	0.5	0.1	1 pixel
Hard X-Ray Photometer	0.7	1	0.3	X-Rays 10 KeV to 150 KeV electrons 300 KeV to 2 MeV protons 1 MeV to 30 MeV includes collimator
shared DPU	3	3		
Sub-Total	17.4	15.2	3.1	
Contingency	3 . - — -7$ 2– . . - -3"			
Total	20.4	.	3.1	

Table II: Small Solar Probe Augmented Payload

Instrument	Mass (kg)	Power (W)	Bit Rate (kbps)	Comments
Plasma Detector	7	7	2.5+	Ions, electrons, TOF, composition to 20 Kev, 2π ster FOV
Magnetometer	3.0°	1.5	0.5+	'Includes 1.0 kg boom
Plasma Wave	5.75	6	2.0+	30 Hz to 30 MHz, 3-axis, includes 0.75 kg for antenna, broadband
Energetic Ions	7.4	6.5	2.0+	$0.02 < E_e < 1$ MeV, $0.02 < E_p < 10$ MeV $1.0 < E_e < 10$ MeV, 15 MeV $< E_p < 25$ MeV, $50 < E_p < 100$ MeV $G_{ions} = 0.1$, $G_{electrons} = 0.0002$ cm^2 ster, includes stepping motor and composition in superthermal range
Coronal Photometer	0.75	1.0	0.6i-	power for motor, 10 pixels
Hard X-Ray Photometer	0.9	1.2	1.0+	X-Rays 5 KeV to 150 KeV electrons 300 KeV to 2 MeV protons 1 MeV to 30 MeV includes collimator, Xenon detector, lower energy X-Ray threshold
shared DPU	5	5		
Sub-Total	29.8	28.2	8.6+	
Contingency	4.5	4.5		
Total	34.3	32.7	8.6+	

+ The instruments can use as much data rate as they can get; (30 kbps desired).

MISSION DESIGN

It requires a large amount of energy to transfer a Small Solar Probe spacecraft from the Earth's surface to within three solar radii of the Sun's surface. Trajectories that do not employ a gravity assist from the massive planet, Jupiter, are found to require post-launch spacecraft maneuvers of 10 km/s or greater. This is unfeasible.

The flyby of Jupiter is required to have a high asymptotic velocity, with respect to Jupiter, of about 13 km/s. This requires energies of the order of 110-120 km^2/s^2 to be imparted to the spacecraft at its departure from the Earth's vicinity. With the large masses of earlier designs there was no existing launch vehicle that could provide this performance, Past mission designers wore forced to resort to trajectories with additional flybys of Earth and/or Venus in order to achieve such energies. This led to flight times from launch to initial perihelion of up to 8 years, a strong driver in the system design process.

Figure 2 shows the estimated performance of selected launch vehicles in the vicinity of 100 km^2/s^2. For the Delta 7925, with a STAR 30C upper stage, the capability is about 170 kg. This becomes the mass constraint for the flight system design. This trajectory type has many benefits that can be taken advantage of in the design process.

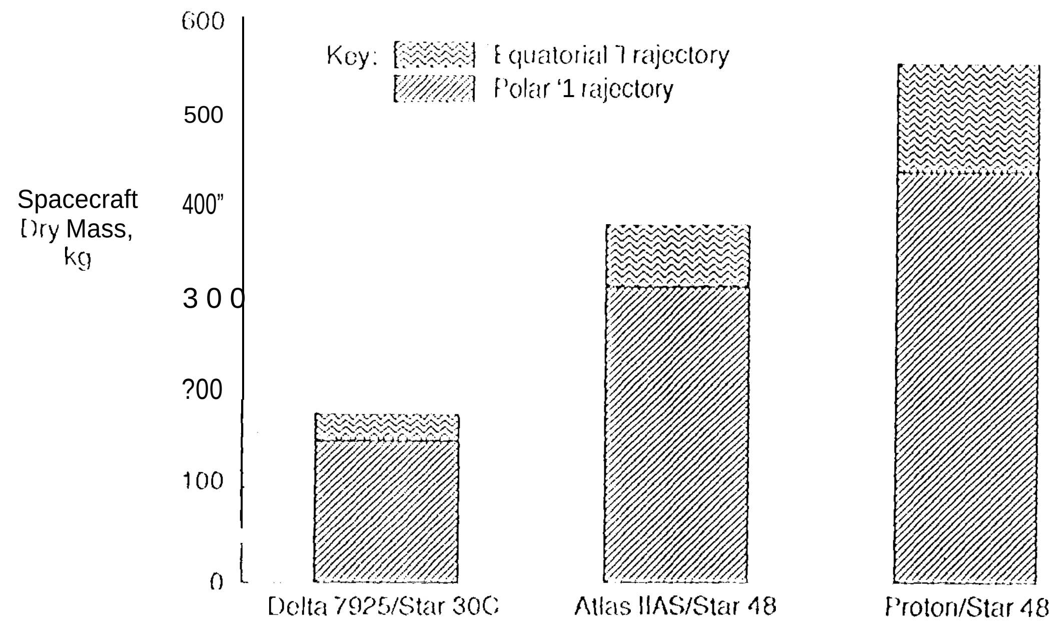

Figure 2: Launch Vehicle Performance for a 10-day Launch Period

1. Launch dates are available every 13 months, the synodic period of Jupiter, and the performance is approximately constant over all launch years. The earlier usage of both Venus and Earth flybys, in addition to Jupiter, in general leads to much longer periods between launch opportunities.
2. The flight time is limited to 3.6 years from launch through the inital perihelion. The earlier designs had durations that reached **8** years implying a higher cost and complexity required for risk mitigation.
3. Propulsion is only required for navigation maneuvers in addition to attitude control. No large deterministic maneuvers are required.
4. The flyby distance at Jupiter is substantially larger than that required in earlier mission 'designs (9.1 R_J cf. $<2 R_J$), thus reducing the radiation exposure of the spacecraft as it passes through the Jupiter system. The environment experienced by the spacecraft is dominated by the conditions very close to the Sun, rather than at Jupiter.

The nominal trajectory is illustrated in Figurer 3 assuming a launch year of 2000. The epoch of perihelion is adjusted to make the Sun-Spacecraft-Earth angle 90 degrees. This requirement is to ensure the maximum angular separation of the Sun and the spacecraft, as seen from the Earth's tracking stations. This will increase the likelihood of successful data transmission through the scintillation effects of the solar corona. It is a science requirement that there be real time data as the spacecraft passes through perihelion.

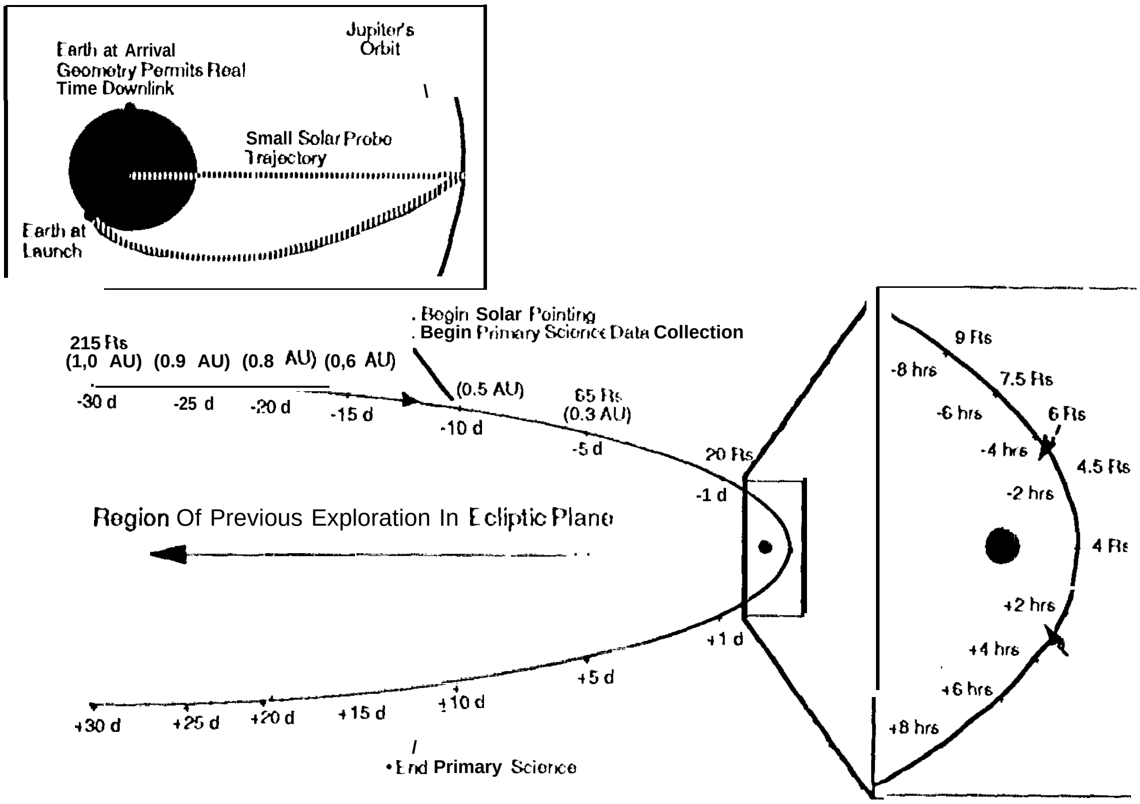

Figure 3: Small Solar Probe Trajectory Profile

This angular requirement actually leads to arrival date choices that am approximately 6 months apart. However it is desired to view, from the Earth, the longitude on the Sun over which the spacecraft will fly prior to perihelion passage of the spacecraft. This leads to a flight time of 3.6 years and to the geometry shown in Figure 3.

FLIGHT SYSTEM DESIGN

Small Solar Probe is an challenging mission for spacecraft design. The prominent features of the mission manifest several driving design requirements for the flight system. The solar flux varies from 0.005 W/cm^2 at Jupiter to 390 W/cm^2 at 3 solar radii from the Sun's surface, exceeding 2800 "suns". T-his extreme range in thermal environment drives the spacecraft configuration as well as the power system. The need to transmit science data real-time during the perihelion pass has dramatic effects on the communications system, electric power system, and spacecraft configuration. Potential rings of dust near perihelion impact the attitude control and communications systems. Charged particle radiation near Jupiter and the Sun pose a threat to spacecraft electronics.

In recent studies the spacecraft design has changed dramatically to reduce the life cycle costs. Most notably, the spacecraft is much smaller than previous designs (Figure 4). Past studies of Solar Probe[3] produced spacecraft designs which were quite large (4 m in diameter and over 16 m tall) and a mass of over 3300 kg. The current Small Solar Probe studies are examining much smaller designs, measuring only one meter in diameter and 4 meters tall. This smaller size contributes to lower costs for spacecraft design, fabrication, testing, operations, and launch systems.

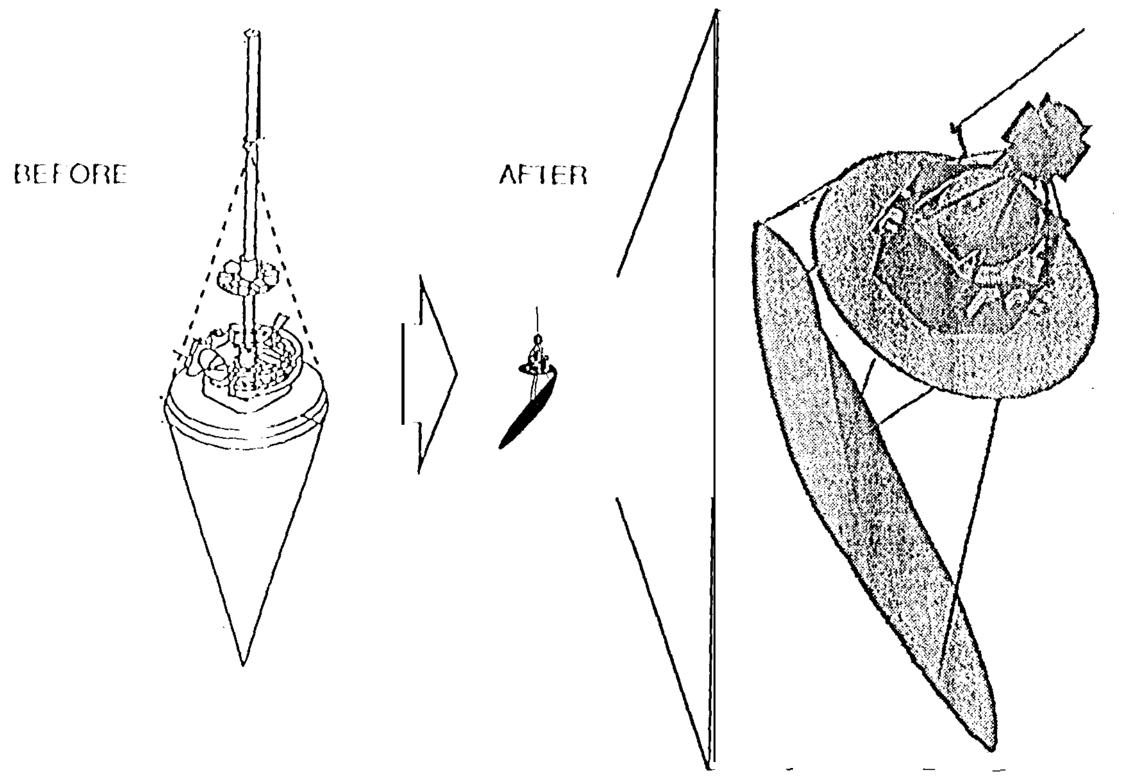

Figure 4: Small Solar Probe is Significantly Smaller than Previous Solar Probe Designs.

The spacecraft design changes visible in Figure 4 are enabled by changes to several requirements which stern from changes in the mission design. The table below presents some of the derived mission and spacecraft design requirements, all of which ultimately flow from the primary requirement to meet a $400M cap on life cycle costs.

Spacecraft Design

The small spacecraft shown is compatible with launch on the Delta II launch vehicle. This spacecraft is approximately one meter in diameter and 4 meters tall with a total launch mass less than 170 kg. The most notable feature of the spacecraft design is the large heat shield which is used to protect the spacecraft from the solar flux during perihelion passage. During perihelion the vehicle is oriented to keep all spacecraft systems within the umbra generated by the heat shield. The boundary condition for the secondary shield design will expose the spacecraft systems to a peak temperature of +30°C (approximately room temperature).

AREA	OLD REQUIREMENT	NEW REQUIREMENT	EFFECTS OF CHANGE
Launch Vehicle	Titan IV / SRMU / Centaur	Delta II 7925 / Star 30C	• Very low LV cost • Small launch mass allowed • Small S/C re uired
Trajectory	VEEJGA or ΔV-EJGA	JGA	• Shorter flight time • No deep space maneuver • No large propulsion system
JCA	3-6 Rj	8-10 Rj	• Lower proton dose at Jupiter
Science Mass	133 kg	< 20 kg	• Smaller S/C permitted
Science Power	103 W	< 20 W	• Smaller S/C power source
Science Data Rate	70 kbps	4 kbps goal	• Less Wer required
Science	Yes	No	• Low mass science instruments
Science Accommodation	Spinning Science Platform	No Spinning Platform	• Less COSI and risk • More mass for science instruments • Multiple sensors for some instruments
	Retractable Magnetometer Boom	Fixed Boom	• Less cost and risk • Less complexity
Number of Perihelion Passes	2	1	• Less complexity • Less fault tolerance • Lower mass and cost
Perihelion Burn Maneuver	Yes, reduce period to ?.5 years	None	• No large propulsion system to shield • Smaller S/C, less risk
Flight Time to First Perihelion	~8 years	3.6 years	• Less complexity • Improved reliability • Lower mass and cost
Mission Lifetime	0+ years	3.6+ years	• Less complexity • Less fault tolerance • Lower mass and cost

Table III. Spacecraft Requirements Changes Help to Lower Total Mission cost.

The Delta launch vehicle has a dramatic effect on the spacecraft design, requiring both low mass and small size. The resultant small umbra volume further restricts the space available for spacecraft components. Reductions in mass and power for some components is achieved by drawing from the technology development program currently under way for the Pluto Fast Flyby mission, specifically in the power and communications subsystems.

Solar Heat Shield

The heat shield shown doubles as a high gain antenna (HGA) reflector to improve communications performance. like the conical design in the earlier configuration, the shield is fabricated from carbon-carbon. The primary design requirement is to limit the total mass loss rate from the shield to less than 2.5 mg/sec in order to limit interference with science measurements. Construction of the 60 mil thick heat shield is very similar to that for the original conical shield design described in Reference 4.

Communications

An all X-band communication system is used to reduce development costs and ensure multiple ground station availability. 1 he small spacecraft size makes it very difficult to place a large high gain antenna within the umbra volume. For this reason, the heat shield is designed to function as a large off-axis reflector to provide over 40 dt3 gain and help reduce dc power requirements. The large gain also helps to combat the scintillation noise expected at X-band. In addition to thermal noise, the communications link will experience phase and amplitude scintillations due to the coronal environment. A real-time downlink is planned for 48 hours at closest approach so as to mitigate the risks associated with the unknown environment. No uplink is planned for this period.

Attitude Control

A cold gas (GN_2) reaction control system is used to provide 3-axis stabilization throughout the mission. During cruise the HGA is pointed toward Earth. In the region within 0.5 AU from the Sun, the heat shield is kept toward the Sun to protect the spacecraft. In this region the spacecraft has limited freedom to move and point the fixed HGA to Earth. Full motion about the yaw axis is possible, however, there is only limited pitch/roll freedom within the time-varying volume of the shield umbra.

1 he solar pressure torque generated by the paraboloidal shield is countered using small hydrazine thrusters. Alternative shield designs are being studied which reduce the solar pressure torque. Light weight star trackers and inertial reference units (IRU) are used for attitude determination. Dust particles are conjectured to exist in the region around perihelion and may interfere with the operation of celestial sensors. I-his possibility drives the IRU selection in order to reduce reliance on celestial sensors at closest approach.

Electronics

A centralized architecture performs command, data handling, and attitude control functions. A shared data processing unit is used for processing, compression, and formating of data from all science instruments. The primary power source is a small RTG (18 kg, 90+W BOM). This is the same unit being developed for the Pluto Fast Flyby mission. Designing the power system to use this RTG provides a large cost savings. Supplemental power during the perihelion pass (P324 hours) is provided by a small primary battery. The new trajectory reduces the expected charged particle radiation dose which helps reduce the mass required for shielding.

Propulsion

1 he trajectory chosen (JGA) eliminates the need for large deep space maneuvers thus eliminating the need for a large propulsion system. A small hydrazine system provides

200 m/s ΔV capability for trajectory correction maneuvers. The tank pressurant is used for attitude control. This integrated design contributes to mass and volume savings.

Structures and Mechanisms
The simple mechanical design has no moving parts, with the exception of the magnetometer boom deployment mechanism. Thermal loads on the heat shield are not expected to appreciably distort the antenna pattern at X-band.

Thermal Control System
The thermal control system must function both in the cold of deep space and at room temperature. All spacecraft systems are blanketed with multi-layer insulation and louvers are used to moderate temperatures.

SUMMARY

Over the past 24 months NASA and JPL have been defining a concept for a Small Solar Probe that is feasible in a technical and fiscal sense, and yet also returns significant science data. It is currently being further defined by two aerospace companies under contract to NASA/JPL. The mission is faster than previous concepts both in its development phase requirements of about 36 months and in its mission duration of 3.6 years from launch to perihelion. The concept is smaller by design of the flight system to be less than 170 kg, the capability of the smaller launch vehicle, the Delta 7925/Star 30C. The cost of the mission is significantly lowered from well over $2 billion to less than $400M. The design is better than earlier concepts by its affordability, its early return of the highest priority science data, and its synergism with other programs, such as that of the Pluto Fast F lyby mission. All lead toward a mission that will intrigue the general public and the educational world by penetrating almost to the heart of our Solar System, the Sun.

ACKNOWLEDGMENT

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